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Assessment of the NASA Glenn 8- by 6-Foot Supersonic Wind Tunnel Supersonic Test Section for Sonic Boom and Supersonic Testing

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August 2022

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Abstract

The NASA Glenn Research Center performed a Sonic Boom Exploratory test with Boeing as an industry partner. The test was performed in the transonic test section of the 8- by 6-Foot Supersonic Wind Tunnel (8x6 SWT) in 2012. Since that test, research has been performed to validate the suitability of the supersonic test section, located upstream of the 8x6 SWT transonic test section. This little used upstream supersonic test section can be the preferred test section for certain objectives in supersonic testing and sonic boom validation. Background pressure profiles were collected in a Mach number stability test (in 2014) with a sonic boom pressure measurement rail installed in the ceiling of the smooth supersonic test section. Results from this test demonstrated favorable background pressure profiles for future sonic boom validation tests. A preliminary supersonic test section calibration was performed in 2016, followed by a detailed calibration in 2020; which also demonstrated the stable and uniform flow characteristics in the supersonic test section. The calibration test utilized the transonic array and the cone-cylinder calibration model in the smooth supersonic test section of the 8x6 SWT. Completion of the calibration prepared the facility for sonic boom assessments of the Low Boom Flight Demonstrator.

Symbols

M_∞	Free-stream Mach number.
P_{to}	Freestream total pressure, psia.
P_s	Static pressure, psia
P_{tb}	Bellmouth total pressure, psia
P_{rail}	Rail static pressure measurement, psia
P_∞	Freestream static pressure, psia
P_t	Total pressure, psia
x_{ts}	Axial distance, referenced from front of test section

1.0 Introduction

NASA is developing technology to reduce the annoyance of supersonic overland flight and to enable public supersonic air travel. An important step in this research was to develop the X–59 Quiet Super Sonic Transport (QUESST) technology demonstrator, an experimental aircraft (x-plane). To support QUESST, wind tunnel assessments of the vehicle sonic boom signature may be required. This report outlines the multi-year effort to assess the NASA Glenn Research Center 8- by 6-Foot Supersonic Wind Tunnel (Ref. 1) (8x6 SWT) in Cleveland, Ohio for reliable sonic boom data.

Historically there is a large body of calibration data (Refs. 2 to 6) for the transonic test section in the 8x6 SWT, but not the supersonic test section. The supersonic and transonic test sections are shown in Figure 1, where the key differences the test sections are (1) the upstream supersonic test section is a smooth-wall test section, and (2) the transonic test section has 4,700 bleed holes to reduce shock reflections. The NASA Glenn Research Center performed a Sonic Boom Exploratory test (Ref. 7) with Boeing as an industry partner. The test was performed in the transonic test section of the 8-by 6-Foot Supersonic Wind Tunnel (8x6 SWT) in 2012. Since that test, research has been performed to validate the suitability of the supersonic test section upstream of the 8x6 SWT transonic test section. This little used upstream supersonic test section can be the preferred test section for certain objectives in supersonic testing and sonic boom validation. Background pressure profiles were collected in a Mach number stability test (in 2014) with a sonic boom pressure measurement rail installed in the ceiling of the smooth supersonic test section. Results from this test demonstrated favorable background pressure profiles for future sonic boom validation tests. A 2-day supersonic test section calibration was performed (in 2016) using the 8x6 SWT transonic array, which also demonstrated the flow characteristics in the supersonic test section. A final calibration (in 2020) test utilized the transonic array and the cone-cylinder calibration model in the smooth supersonic test section of the 8x6 SWT. Completion of the calibration prepared the facility for sonic boom assessments of the X–59 (QueSST) Low Boom Flight Demonstrator.

The subject of this report is to present the data from the Sonic Boom Exploratory test, the Mach Number Stability test, and the supersonic test section calibrations. Results will be presented from (1) the sonic boom pressure rail as installed in the transonic test section, (2) the sonic boom pressure rail as installed in the supersonic test section, and (3) the total pressure and static pressure calibration array as installed in the supersonic test section; with comparison to Computational Fluid Dynamics (CFD) calculations.

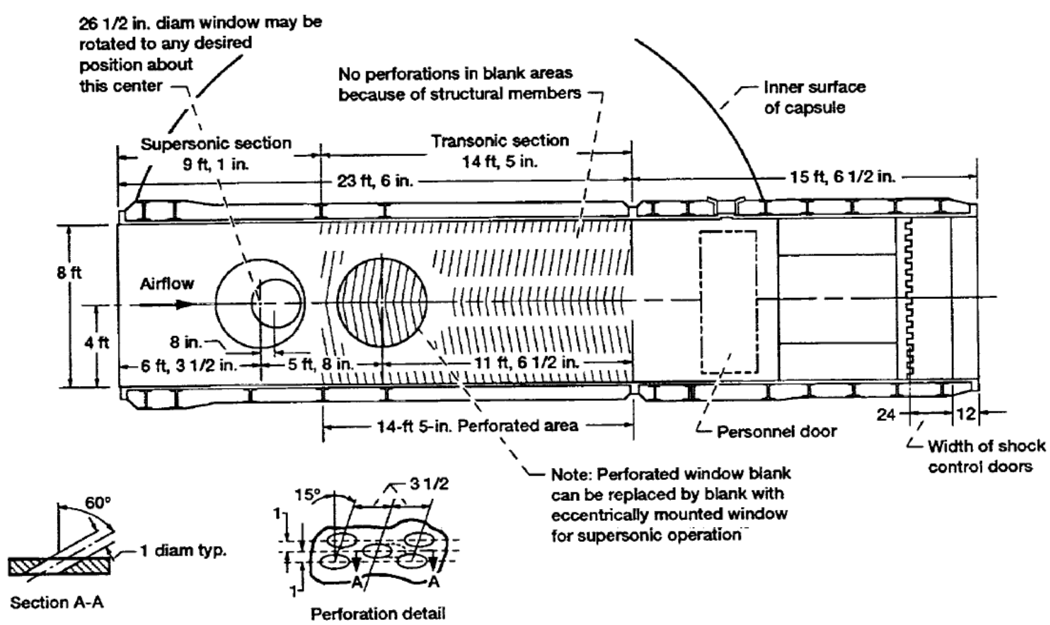

Figure 1.—The 8x6 SWT Test Section Layout, dimension in inches.

2.0 Test Setup

2.1 Sonic Boom Exploratory Test (2012)

The 8x6 SWT transonic test section was configured with the 14-ft test section, setup for Schlieren imaging, and the 5.8 percent porosity setting, shown in Figure 1. Porous blanks, where the bleed holes were filled with plugs, were installed in the upstream window openings. The Schlieren windows were rotated to the lower 45° position. Halfway through the test there were concerns regarding the porous blanks influencing data quality on the pressure rail. The Schlieren windows were re-located to the upstream position and the porous blanks downstream. Test section configuration after this switch was the 14-ft test section and 5.8 percent modified porosity setting. The Schlieren windows were rotated so that they were positioned upstream and on the tunnel centerline. The window position did not affect the results.

Tunnel instrumentation includes wind tunnel inlet bell mouth total pressure rakes and an array of ceiling static pressure taps.

Three models were tested at supersonic speeds ranging from Mach 1.2 to 1.8. A pressure rail was installed in ceiling plates along the centerline of the test section to capture model pressure signature data (Figure 2 and Figure 3). The model was installed on the sting in an upside-down orientation, so the pressure signature, or boom signature, from the bottom of the model was measured by the pressure rail. A force balance was installed between the model and the strut and was used to measure both forces and moments on the model; and to determine model deflections.

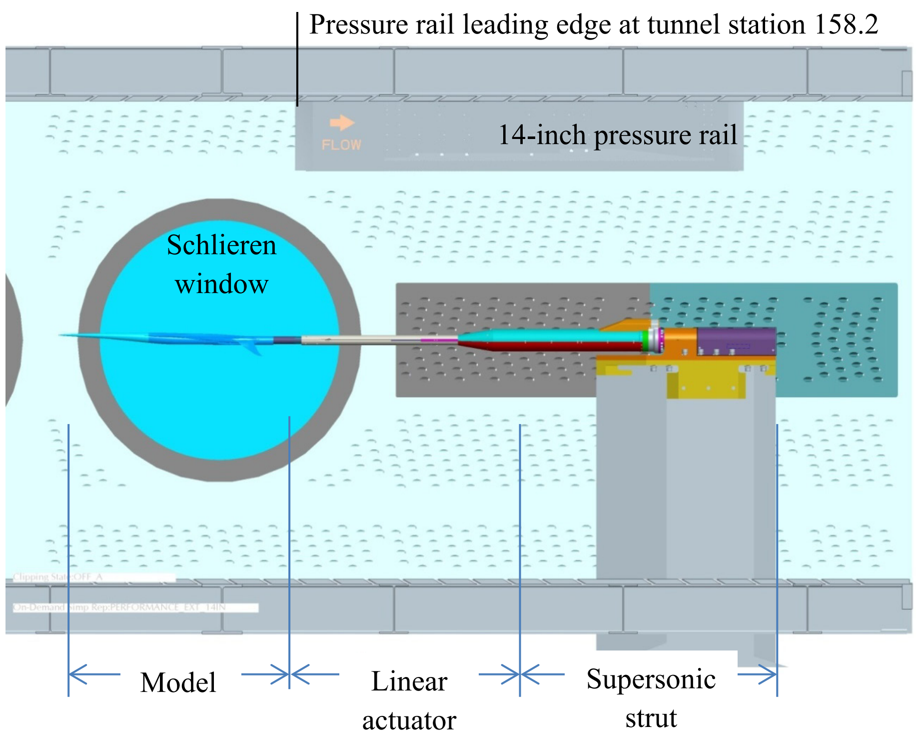

Figure 2.—Layout of the Boeing Sonic Boom Exploratory Test, 8x6 SWT Transonic Test Section, 2012.

Figure 3.—Photo of the Boeing Sonic Boom Exploratory Test, 8x6 SWT Transonic Test Section, 2012.

During the test, each model's spatial position was varied horizontally (X-axis, in line with the sting) up to 24 in. by a linear actuator and vertically (Z-axis) up to 64 in. by raising and lowering the supersonic strut. Pressure measurements from the translations were averaged to generate the vehicle pressure profiles. The supersonic strut position was controlled remotely and was raised or lowered while the tunnel was operating.

The sonic boom pressure rail was 90 in. long, 14 in. tall, and 1 in. thick at its base. The rail tapered down to 0.050-in. thickness at the tip. Instrumentation included 420 pressure taps distributed 0.1575 in. apart over a 66-in. length, Figure 4. The leading edge of the pressure rail was positioned at Tunnel Station 158.2.

Spatial averaging was conducted by averaging data at several positions in the fore and aft direction for an "X-sweep", or in the vertical direction for a "Z-sweep". The model was moved in incremental distances of approximately 4 rail pressure ports, for a total of 13 model positions.

Test conditions were nominally between Mach 1.2 and 1.8, where Mach number was initially held to ±0.01, and later improved to ±0.001. In the 8x6 SWT, Mach number is controlled by three variables: (1) flex wall nozzle throat position, (2) compressor speed, and (3) balance chamber exhaust valve position. The exhaust valve in the balance chamber controls the flow of air exiting the porous walls, which influences the static pressure in the test section.

Measurements of vehicle pressure profile ($\Delta P/P$) were collected as defined by the following equation:

$$\Delta P/P = ((P_{rail} - P_{\infty})/P_{\infty})_{model} - ((P_{rail} - P_{\infty})/P_{\infty})_{reference\ run}$$

2.2 Mach Number Stability Test (2014)

Measurements were taken in the supersonic test section, and for documentation purposes the 8x6 SWT transonic test section was again configured with the 14-ft test section, setup for Schlieren imaging, and the 5.8 percent porosity setting. No model was installed. The supersonic strut was installed both flush with the floor and moved to a variety of strut heights.

The sonic boom pressure rail was again used this time in the supersonic test section. The leading edge of the pressure rail (Figure 4 and Figure 5) was positioned at Tunnel Station 10.96 in.

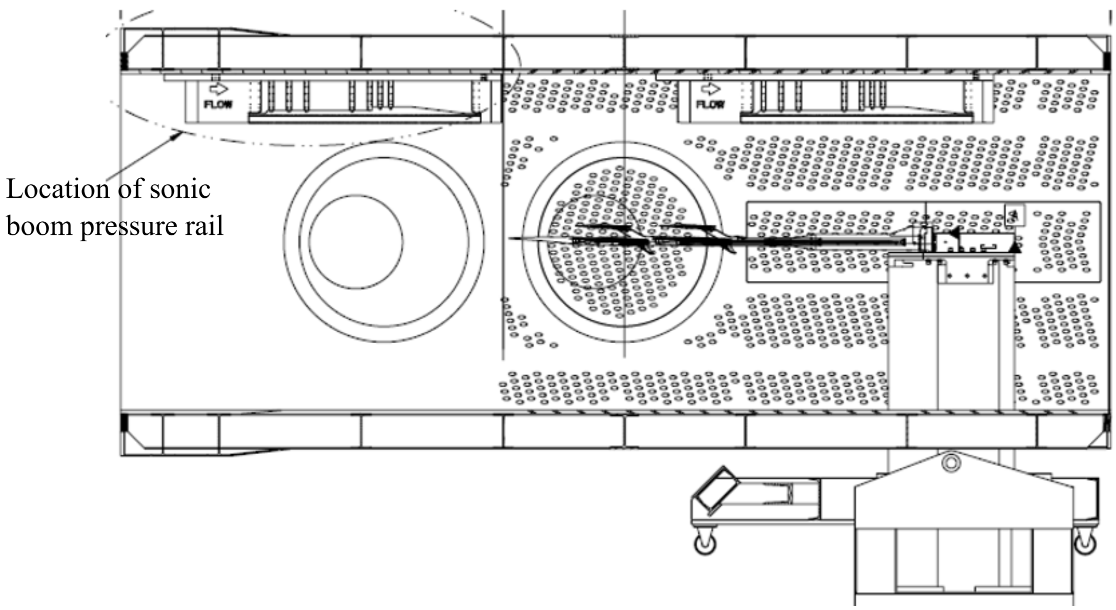

Figure 4.—Layout of rail locations for the Mach Number Stability test, 2014. Upstream location was for the 2014 Mach stability test and downstream location was for the 2012 Boeing Sonic Boom Exploratory Test.

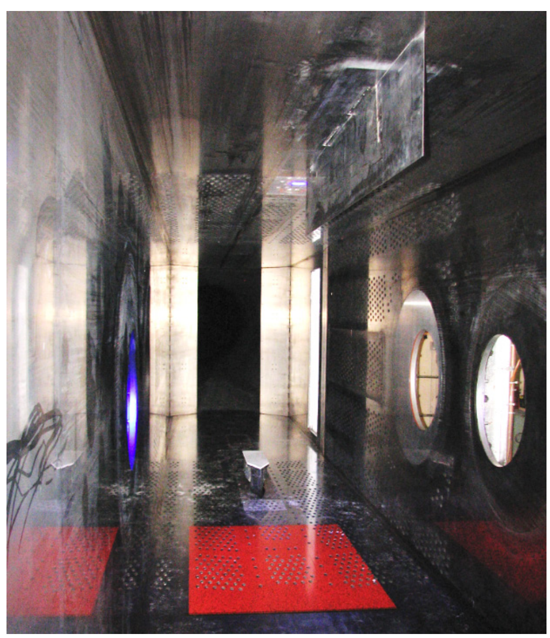

Figure 5.—Pressure rail installed in the Supersonic Test Section, 2014.

2.3 Supersonic Test Section Calibration (2016 and 2020)

Measurements were taken in the supersonic test section, and again for documentation purposes the 8x6 SWT transonic test section was again configured with the 14-ft test section, setup for Schlieren imaging, and the 5.8 percent porosity setting, shown in Figure 1. No model was installed. An array of total pressure and static pressure instrumentation was installed (Figure 7), called the “transonic array”.

The array instrumentation is typically comprised of 5 flow angularity probes, 6 pitot-static pressure probes, 11 thermocouples and two hot film anemometry probes (Figure 6 and Figure 7). The array is typically sting-mounted to the transonic strut and is further supported by wall plates attached to both ends of the array body and by a vertical support downstream of the array body. The porous Schlieren window blanks, while in the upstream location, allow the array’s wall plates to be mounted at centerline or ±1-ft from centerline.

The flow angle probes are a five-hole, hemispherical-head design that allows for the resolution of two flow angle components (pitch and yaw). The flow angle probes were calibrated for a Mach number range of 0.5 to 2.0 (the Mach number range of the 8x6 SWT is 0.36 to 2.0 for three motor operation; the Mach number range of the probe calibration facility located at Sandia Labs was 0.5 to 4.0 so that probe calibration data was not obtained corresponding to the low end of the 8x6 SWT Mach number range; at these low Mach number conditions, probe calibration parameters are extrapolated from the calibration data). These probes extend 21.125 in. from the leading edge of the array. The pitot-static probes also extend 21.125 in. from the array leading edge. The thermocouples are mounted to the bottom of the array body, with the heads of the thermocouples about even with the array leading edge.

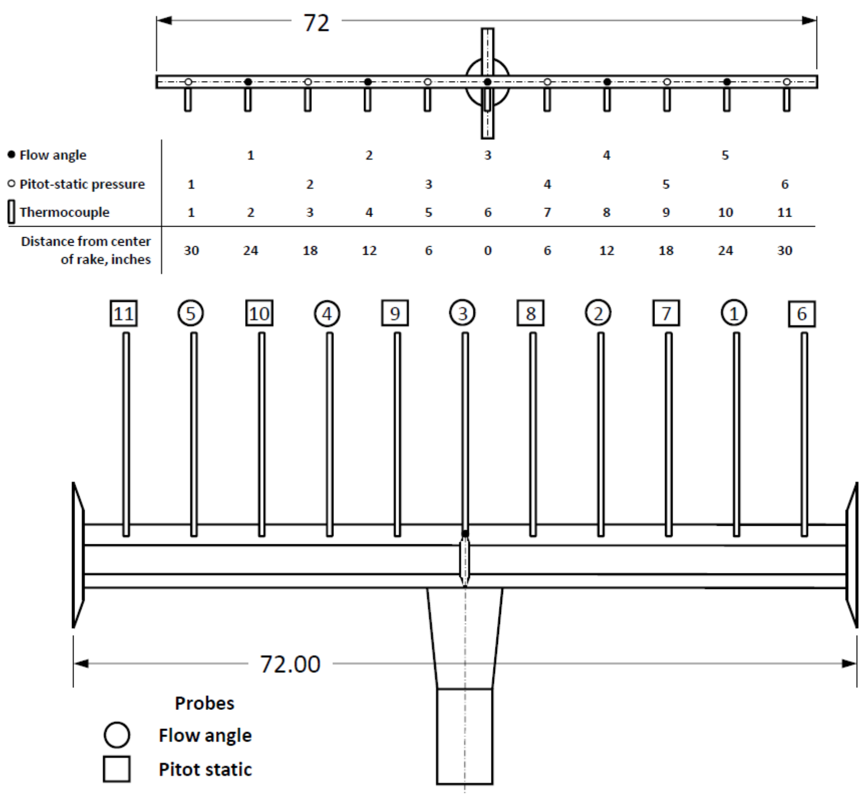

Figure 6.—Total pressure and static pressure measurement array.

Figure 7.—Total pressure and static pressure measurement array and boundary layer rakes installed in the 8x6 SWT Supersonic Test Section.

Boundary layer rakes were also installed on the wind tunnel floor and ceiling, in nearly the same plane as the transonic array (tunnel station 3.6875 for transonic array versus tunnel station 3.75 for boundary layer rakes).

Details on the calibration performed in 2020, performed similarly to the calibration performed in 2016, are provided in Reference 11.

3.0 Wind-US CFD Code

The empty tunnel 8x6 SWT was modeled with WIND-US, a general-purpose fluid flow solver (Ref. 8). WIND-US was used to take advantage of the established capability to correctly compute wind tunnel flow with viscous and turbulence effects. The code supports the solution of the Euler and Navier-Stokes equations, along with supporting equation sets governing turbulent and chemically reacting flows. The flow solver is parallelized and can take advantage of multi-core and multi-CPU hardware. The version used was WIND-US version 4.6.

WIND-US was used with the modified second-order Roe upwind scheme for stretched grids, implicit time stepping with a Courant–Friedrichs–Lewy (CFL) number of 1.0, and the Menter Shear Stress Transport (SST) turbulence model. The 8x6 SWT was modeled from the plenum upstream of the tunnel bellmouth, to the diffuser entrance. A structured grid was used with viscous walls and spacing of the first grid point from the wall at $y^+ = 1.0$. A grid sensitivity study was performed but is not summarized in this report. Individual bleed holes were not modeled but a uniform bleed model was used for the walls in the 8x6 SWT transonic test section. Convergence was monitored using numerical residuals and wind tunnel mass flow variation of less than 0.25 percent.

4.0 Results

4.1 Sonic Boom Exploratory Test (2012)

The sonic boom pressure profile, or signature, was initially difficult to obtain in the transonic test section. Signatures were characterized by a large zero offset and large variation from the predicted signature. Signatures were collected by moving the model in small axial increments when positioned below the rail, taking data at each location and averaging the results (Ref. 9). A background rail pressure signature was then subtracted from these averaged results. When in the 8x6 SWT transonic test section, the background pressure profile (Figure 8) was found to have a large amount of spatial variation, where the $\Delta P/P$ varied in some locations from –0.06 to 0.06. The background signature also had regions of steep $\Delta P/P$ gradients. During the first portion of the exploratory test the stability of the Mach number was ±0.01; as this tunnel Mach number drifted, the steep gradients in the background pressure profile would move spatially in the wind tunnel. This movement or drift in the background profile was the major cause of zero offset and variation from the predicted signature. The drift was traced to the operation of the tunnel balance chamber exhaust valve which controls the flow exiting the tunnel bleed holes. Operating the valve in a manual mode held the Mach number to within ±0.001, which greatly improved the sonic boom signatures collected during the test (Ref. 9).

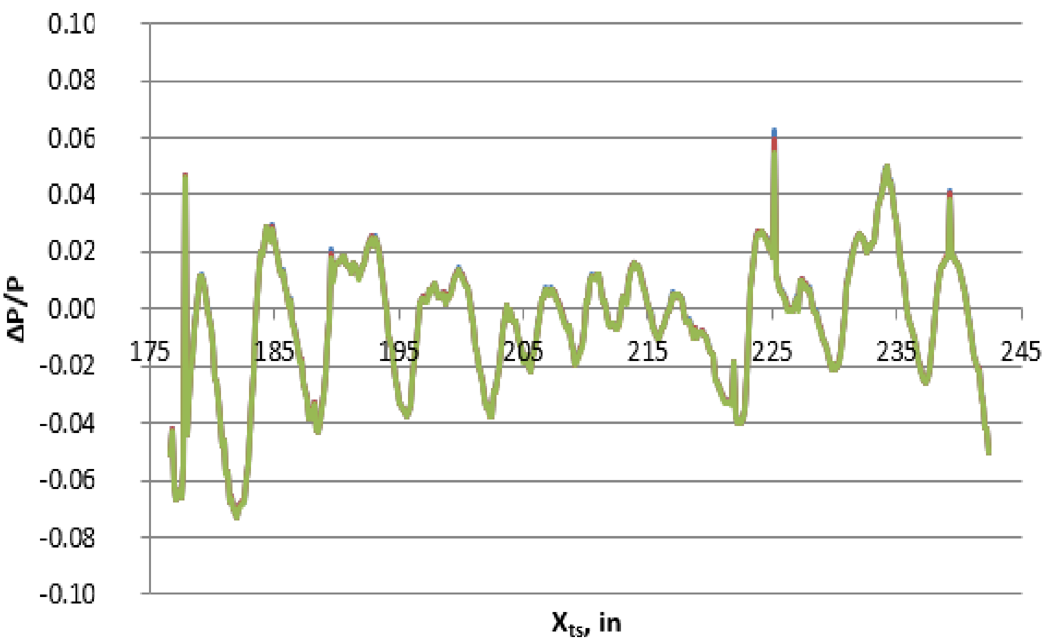

Figure 8.—Mach 1.6 Background Pressure Profile from 8x6 SWT Exploratory Test, 2012.

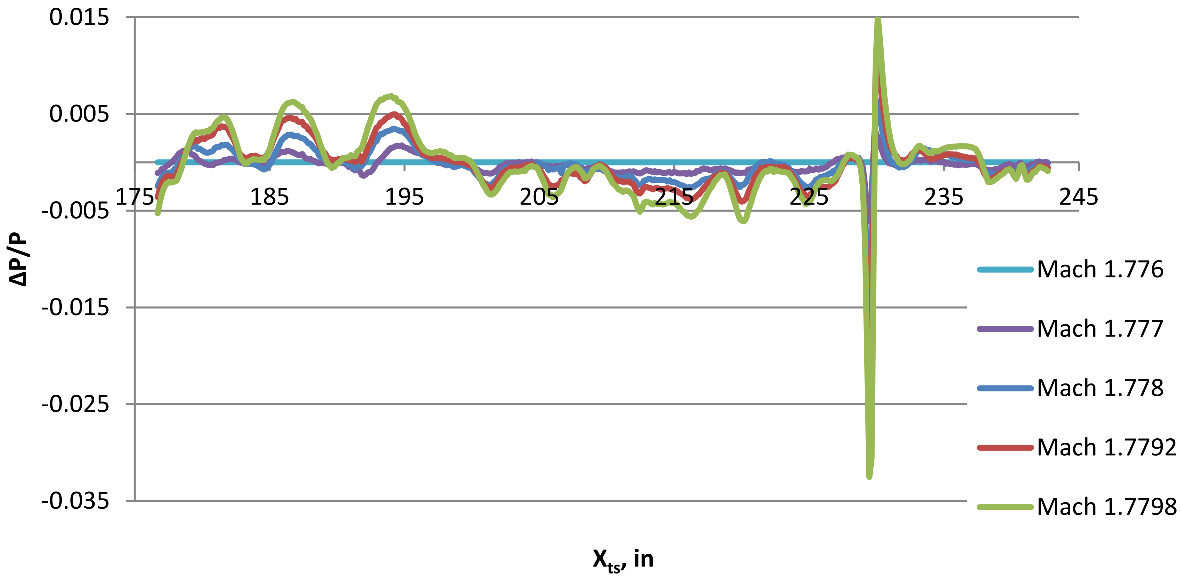

Figure 9.—Empty tunnel Ps stability (variation with Mach number) as measured by pressure rail, transonic test section, Exploratory Test, 2012.

Figure 9 demonstrates how the variation of the Mach number created disturbances in the measured ΔP/P; data in the figures is ΔP/P independently averaged at Mach 1.776, 1.777, 1.778, 1.779, and 1.780. For each Mach number tested the background ΔP/P (independently collected at Mach 1.776) was subtracted. As the Mach number increased each increment of 0.001, the variation in measurement at many locations in the signature increased dramatically. For example, at x_{ts} = 194 in. the pressure signature changed from 0 to 0.005. These are the variations that cause the zero offset and variation in the sonic boom signature; and shows the need to maintain Mach number ±0.001 or lower.

4.2 Mach Number Stability Test (2014)

Based on results from the exploratory test, the 8x6 SWT supersonic test section was studied to see if improved background pressure profile measurements could be obtained. No documented calibration data existed for the supersonic test section at that time. The sonic boom pressure rail was installed in the supersonic test section to check the stability of measurements, though the exact Mach number in the test section was unknown at that time.

Figure 10 to Figure 13 show the background pressure signatures from the maximum Mach number of 2.0 down to 1.4. In general, the background variation in ΔP/P was greatly reduced to ±0.02 (was ±0.06 in the transonic test section) for Mach 2.0 and 1.8; and the large number of steep gradients in ΔP/P were absent.

At Mach 1.6, the background signature had smooth variation between x = 45 in. to x = 85 in. (ΔP/P –0.25 to 0.00). At x = 90 in. the rail nose shock reflected off the wind tunnel walls and affects the rail measurements. This high gradient region may need to be avoided for measurements if tunnel Mach number cannot be held to a tight tolerance.

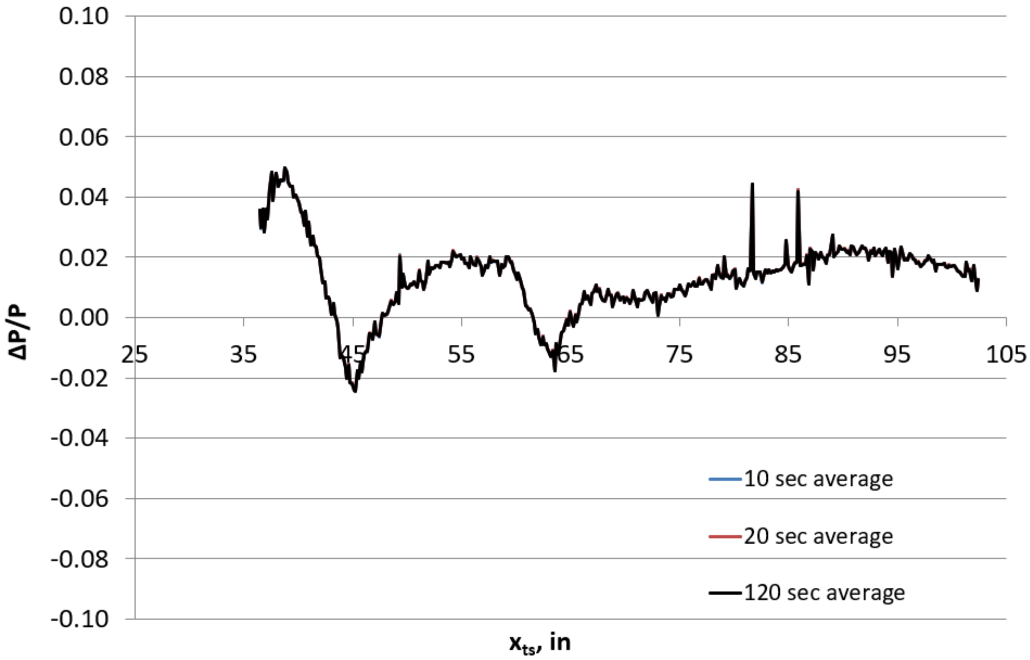

Figure 10.—Mach 2.0 Background pressure profile, Supersonic Test Section, 2014.

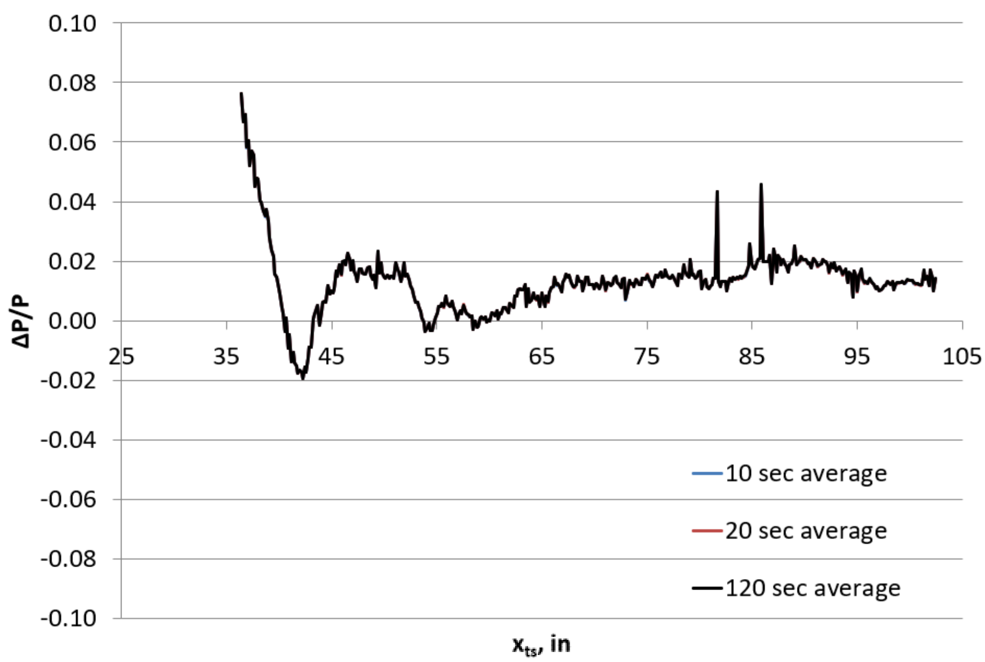

Figure 11.—Mach 1.8 Background pressure profile, Supersonic Test Section, 2014.

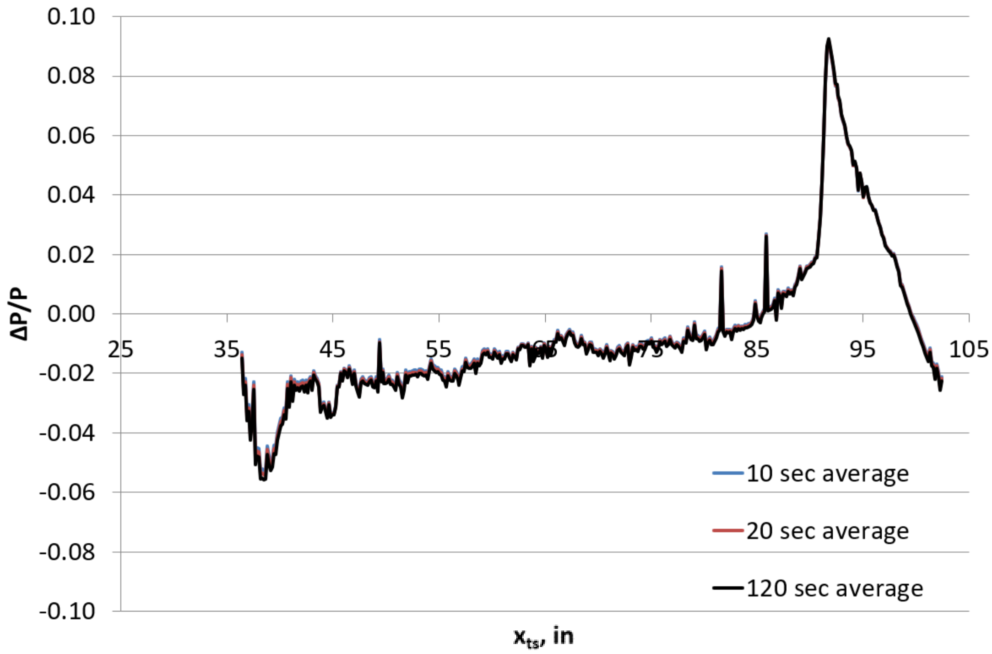

Figure 12.—Mach 1.6 Background pressure profile, Supersonic Test Section, 2014.

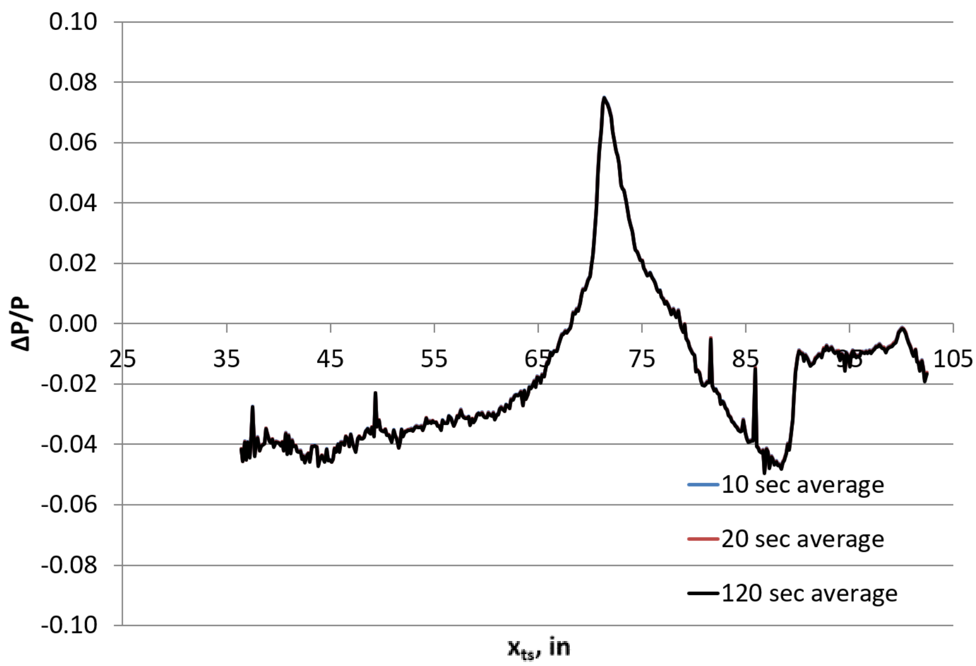

Figure 13.—Mach 1.4 Background pressure profile, Supersonic Test Section, 2014.

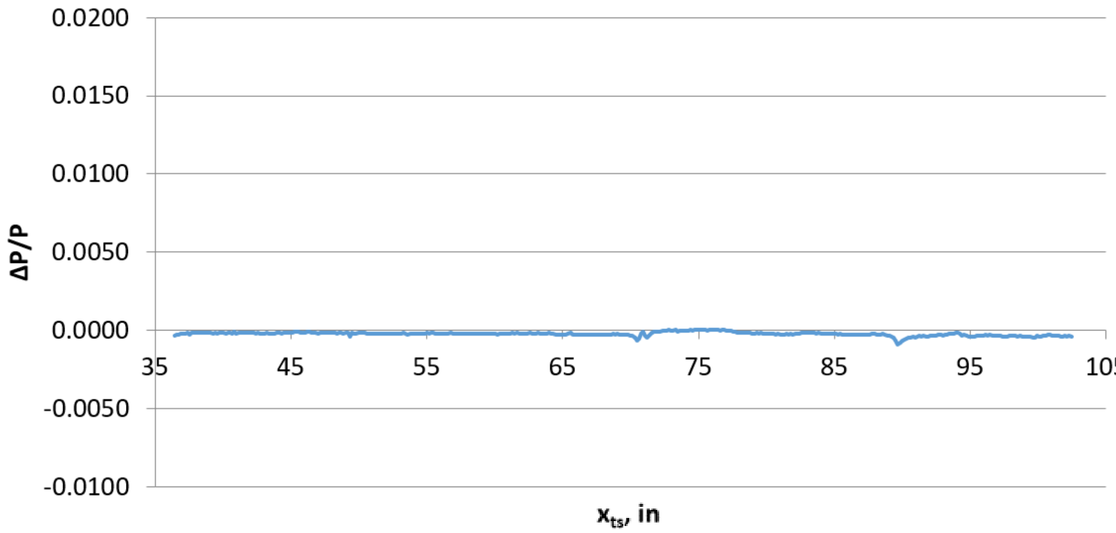

Figure 14.—Empty tunnel Ps stability as measured by pressure rail, 2014. Mach 1.4 Supersonic Test Section, corrected ΔP/P.

At Mach 1.4 results were similar, where ΔP/P varied smoothly from –0.04 to 0.0 between x = 35 in. and 65 in. Much less rail may be available for measurements at this Mach number, again only if wind tunnel Mach number cannot be held to a tight tolerance.

Assuming the Mach number can be held steady with variation better than ±0.001, then the steep pressure gradients will not affect boom measurements as demonstrated in Figure 14. In this figure, two background pressure profiles were collected at two separate points in time and subtracted. Stability of the empty tunnel sonic boom measurement ΔP/P was zero, ±0.001 or better.

Rail results at Mach 1.2 are in Figure 15, the background ΔP/P exhibited large variation due to the reflection of the nose shock from the rail that reflects off the tunnel walls back to the rail. However, there were no regions with steep ΔP/P gradients. If tunnel Mach number is held to ±0.001 or better, then sonic boom measurements would be stable.

4.3 Supersonic Test Section Calibration (2016 and 2020)

Following the Mach stability test, a supersonic test section calibration was performed in 2016; with a more extensive calibration (Ref. 11) performed in 2020. Figure 16 displays the static pressure measured from a row of ports located on the wind tunnel ceiling located 1.5 ft from the tunnel centerline. Measurements between x = 5 and 8 ft were influenced by the presence of the transonic array used to measure tunnel pitot pressure. The ceiling static pressure profile was also computed with CFD and compared to the experimental data in the figure, the transonic array was not included in the empty tunnel CFD. Comparisons demonstrate differences in P_s/P_{tb} ratio between CFD and the experimental data better than ±0.02.

Results are shown for Mach 1.56. Figure 17 show the six static pressure measurements located at an axial station on the ceiling of x = 1.42 ft. At this location the experimental measurements were ±0.003 (P_s/P_{tb} ratio) or better when compared to CFD computation. When averaged, these static pressure ports were suitable for future use as a calibrated static pressure measurement to represent the supersonic test section static pressure. These measurements were closely coupled to the test section and represented an improvement over use of balance chamber pressure. Similarly, the static pressure measurements located on the wind tunnel walls were compared to computations (Figure 18) with agreement ±0.001.

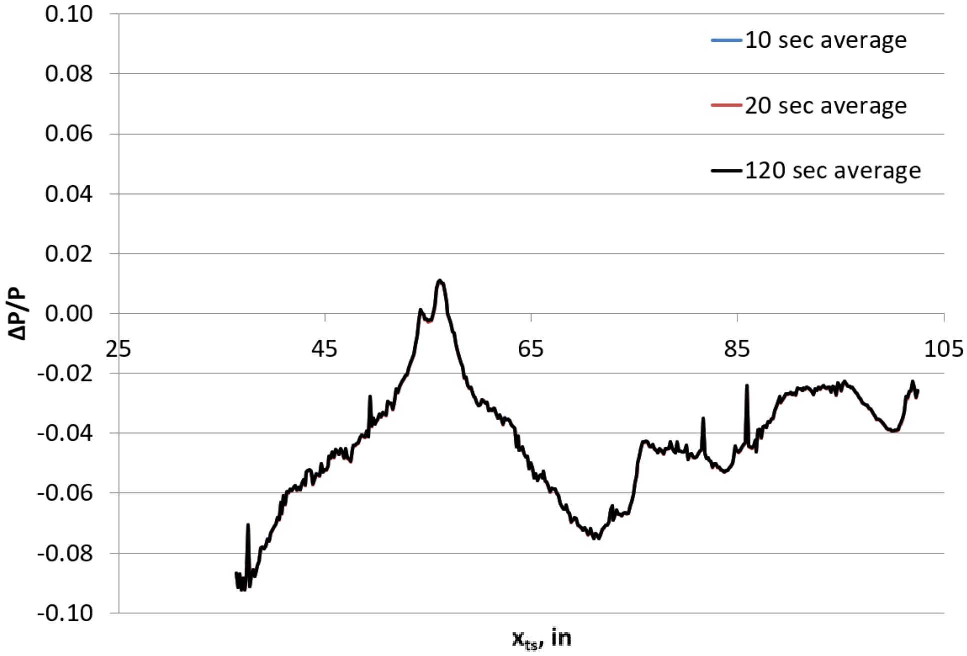

Figure 15.—Mach 1.2 Background pressure profile, Supersonic Test Section, 2014.

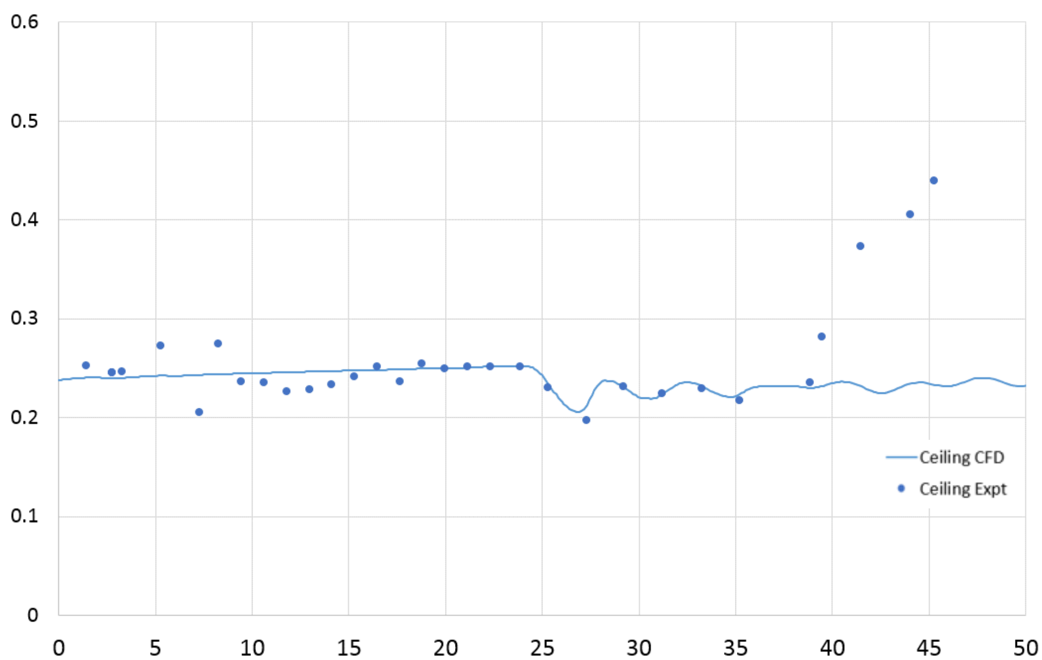

Figure 16.—Mach 1.56 Ceiling Static Pressure Ports (P_s/P_{tb}) vs. x_{ts}, ft.

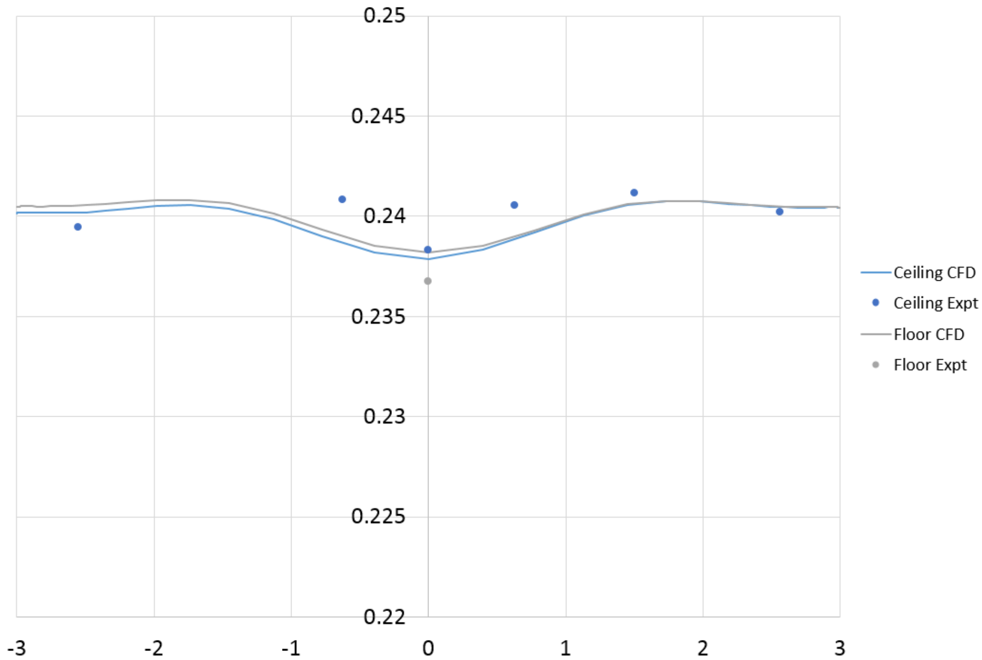

Figure 17.—Mach 1.56 Ceiling and Floor Ports Static Pressure Ratio (P_s/P_{tb}) vs. width, ft.

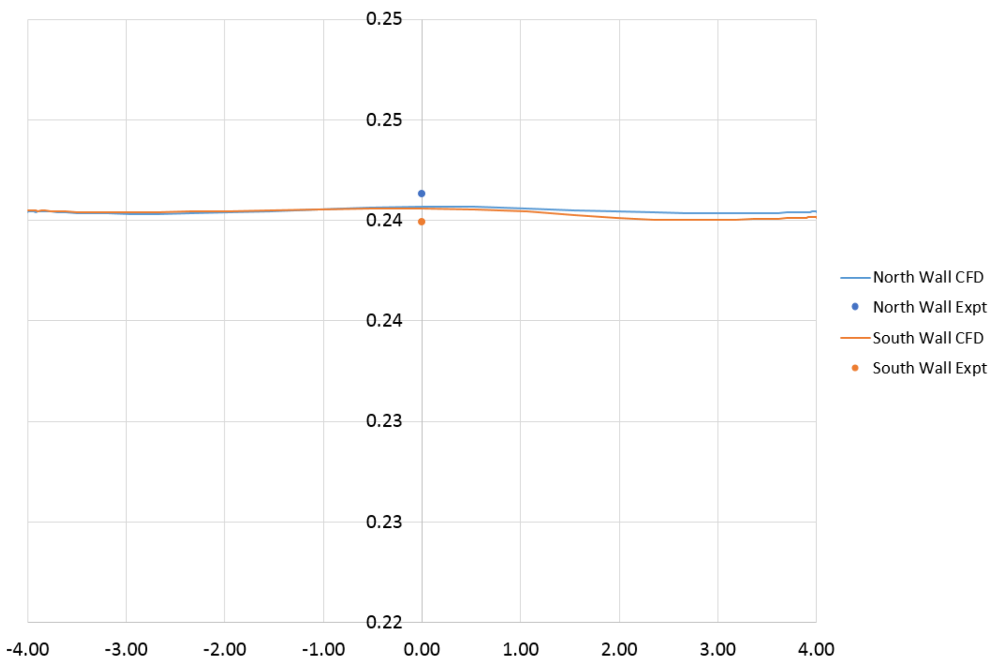

Figure 18.—Mach 1.56 Wall Static Pressure Ports (Ratio P_s/P_{tb}) vs. height, ft.

Plotted in Figure 19 is the pitot pressure as measured by boundary layer rakes located at an axial tunnel location of x = 3.75 ft. Ignoring a couple leaking probe tips between –2.5 to –4.0, CFD predictions agreed with experimental data at a level of ±0.01 (P_t/P_{tb}) or better. Similarly, pitot pressure and static pressure as collected from the transonic array (Figure 20) were collected at the axial tunnel location x = 3.6875 ft. with variation between CFD and data less than ±0.01.

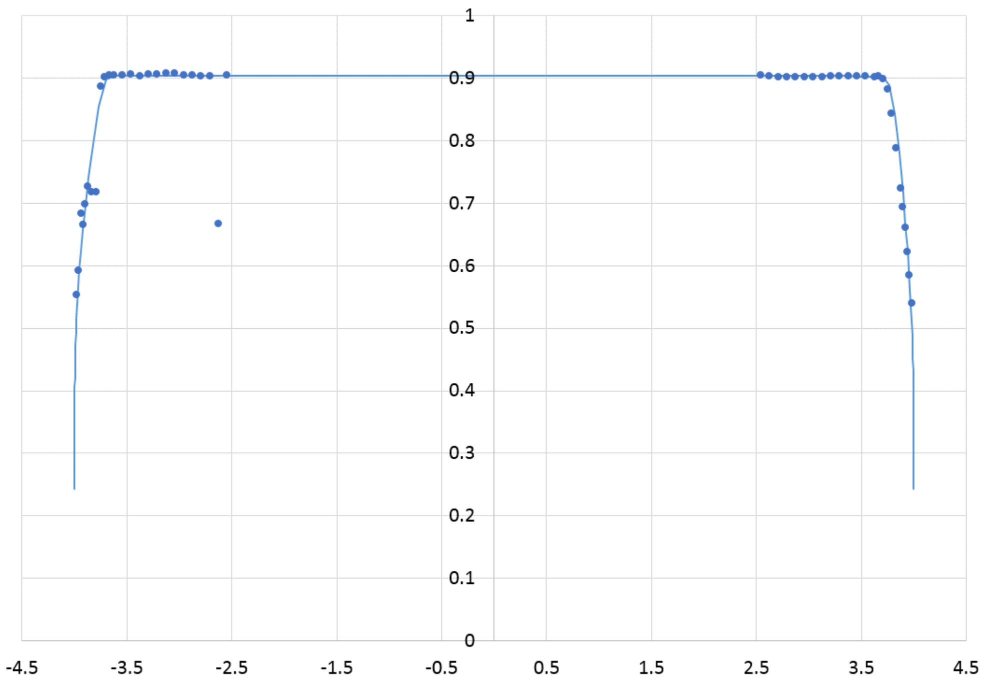

Figure 19.—Mach 1.56 Pitot Pressure Ratio (P_t/P_{tb}) Boundary Layer Rakes vs. height, ft.

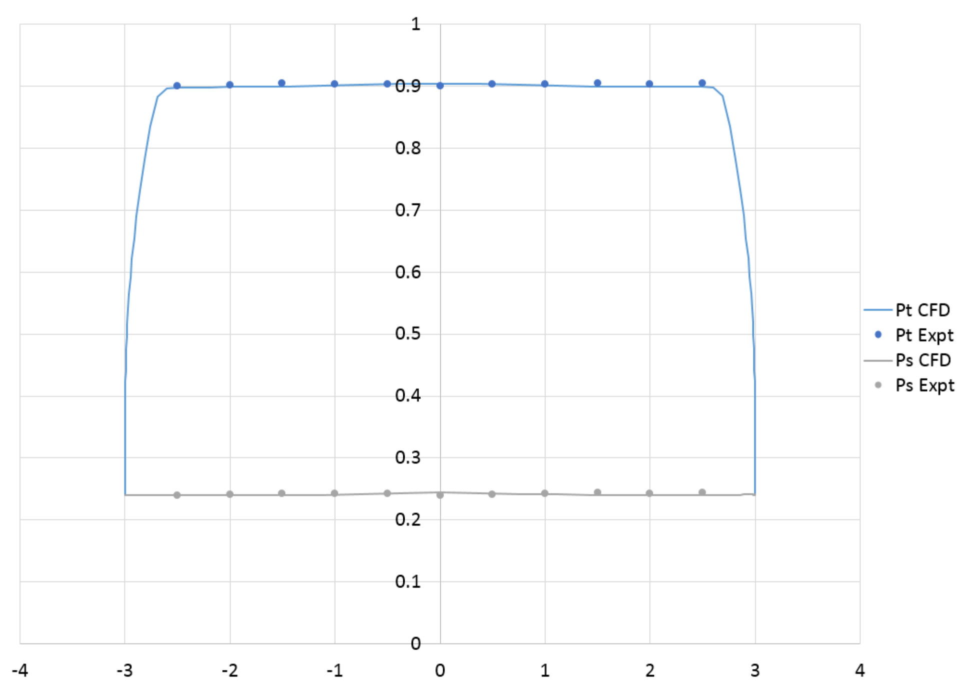

Figure 20.—Mach 1.56 Pitot Pressure Ratio (P_t/P_{tb}) and Static Pressure Ratio (P_s/P_{tb}); Transonic Array vs. width, ft.

Calibration data and CFD were performed at wind tunnel Mach numbers between Mach 1.2 and 1.8. Results were similar at all Mach numbers collected. As a result, this report will only present results obtained from Mach 1.56.

5.0 Summary

This report summarized results from three test activities: the sonic boom exploratory test, the Mach number stability test, and the supersonic test section calibration. These three efforts together created a database to demonstrate the flow quality of the 8x6 SWT transonic and the supersonic test sections for use in sonic boom testing.

Key findings were that the background pressure profiles in the supersonic test section demonstrated much less variation in the pressure signature than the transonic test section. The supersonic test section also had fewer regions with steep gradients in the pressure signature than the transonic test section. Testing also demonstrated the ability of the wind tunnel to hold Mach number to a tolerance better than ±0.001; the corresponding improvements were realized in the background pressure signature measurements.

In the transonic test section, the 8x6 SWT uses an average of four static pressure measurements in the test section balance chamber to compute a calibrated value for the tunnel static pressure. As the balance chamber is a large volume with flow exiting the wind tunnel bleed holes, there is the potential for pressure in the balance chamber to vary differently than the actual static pressure in the test section. Closer coupling of this measurement to the test section could alleviate the problem which also adversely affects sonic boom measurement.

In the supersonic test section rail measurements will be improved if the tunnel Mach number is referenced to a value also collected from the supersonic test section; and more closely coupled to the bell mouth total pressure measurement. In this way variations in compressor speed affect both the total pressure measurement and calibrated static pressure measurement equally, resulting in improved Mach number stability.

As a result, the facility will be ready to collect sonic boom signatures using the sonic boom pressure rail for Low Boom Flight Demonstrator sonic boom assessments and comparison to flight data. Other types of supersonic testing above Mach 1.2 may also benefit from being in the flow field with improved Mach number stability found in the upstream supersonic test section; where types of testing that may benefit include high speed tests with dynamic instrumentation, among others.

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